ALTERNATIVE POWER-GENERATION SYSTEMS

Robert E. English
NASA Lewis Research Center

INTRODUCTION

At present, Earth-orbital power systems consist almost exclusively of photovoltaic arrays and batteries. Because the characteristics of this class of power system are both well known and gradually improving through evolution, mission planners are on familiar ground in selecting photovoltaic power systems. The photovoltaic system, of course, requires orientation of a solar array of large area toward the Sun. This array obscures the field of view, adds to atmospheric drag in low orbit, and could possibly interfere with rendezvous or with departure from an orbiting spacecraft. The performance of the photovoltaic array also degrades as a result of radiation damage, and the batteries used for energy storage are of limited life in low Earth orbit.

Thermal space power systems have very different characteristics. Chiefly, they are more compact, of long life, and far less susceptible to radiation damage than photovoltaic systems. Those power systems that obtain their thermal input from nuclear heat sources can produce power whether in sunlight or shade and without the orientation toward the Sun required by the solar arrays.

Like the photovoltaic power systems, the thermal power systems are also evolving; but, unlike the photovoltaic systems, they have had comparatively little use in space. This paper surveys the present state of the art of thermal power systems. Because of the great potential variety of thermal power systems, the heat sources, the power-conversion systems, and the integration of thermal power systems with missions are treated sequentially.

SOLAR HEAT SOURCES

The Sun emits radiant energy equivalent to that from a blackbody at about 5800 K. At the Earth's distance from the Sun, the Sun's thermal flux is 1400 watts per square meter. Because a paraboloidal mirror and solar heat receiver can collect at least 80 percent of this energy, solar mirrors can provide 10 times the heat per unit of collector area that is obtainable from photovoltaic arrays in combination with a resistance heater. Thus, solar mirrors have a great size advantage over solar-cell arrays if the energy sought is heat.

Inasmuch as a power-conversion efficiency of 0.30 is readily achievable, an electric power output of about 350 watts per square meter is potentially obtainable from a solar thermal power system during full-sun operation - about three times the power from arrays of solar cells. The technologies that can provide these high levels of either power or heat are thus of considerable interest.

A perfect paraboloidal mirror can produce a small image of the Sun, the image size being determined by the mirror's focal length and the Sun's apparent radius of 4.8 milliradians (16 arc-min). On the other hand, a real mirror will have surface inaccuracies and will therefore produce a larger image. The resulting high flux of solar energy can be focused on an aperture in a heat-receiving cavity (a hohlraum, fig. 1), and the thermal energy collected by the heat receiver can be used directly or can be converted to electric power in a thermal power system. The hot cavity will radiate heat through the cavity's aperture as would a black surface at the mean radiant temperature within the cavity. Because geometrical errors in the mirror's surface increase image size and thereby require a larger aperture, these surface errors result in an increase in the thermal power lost by radiation from within the cavity. By exploring the relation among mirror-surface error, aperture size, and radiation loss through the aperture, the efficiency achievable by solar heat-collection systems can be assessed.

For specificity in the following discussion, a mirror is assumed to be a paraboloid of revolution, to have a diameter of 30.5 meters (100 ft), and to be so oriented in space that both the Sun and the aperture of the solar heat receiver are centrally positioned on the axis of the paraboloid. A mirror of this size intercepts roughly 1 megawatt of sunlight. Further, a ray from the mirror lip to the focus is taken to form an angle of 45° with the mirror axis. Under these conditions, focal length is 18.4 meters (60.4 ft) and the f-number of the optical system is f/0.6.

Each area element of the mirror surface forms a circular image of the Sun of 8.6-centimeter radius at the image plane. For surface elements correctly oriented, the Sun's image formed by each element is centered on the mirror's axis. Accordingly, an error in orientation of a given surface element displaces the Sun's image formed by that element radially from its nominal location on the mirror's axis. For analysis, the surface errors were assumed to have a Gaussian distribution, that is,

$$p = \sqrt{\frac{2}{\pi}} \exp\left[-\frac{1}{2}\left(\frac{\varepsilon}{\sigma}\right)^2\right]$$

where

p probability density for a given error

ε surface error

σ standard error

For various radial positions on the image plane, the flux from various elements of the mirror surface was integrated over this probability distribution. The results are given in figure 2.

The perfect image (zero error) has a flux cutoff at the image radius of 8.6 centimeters. As the standard error of the mirror surface increases, the image is spread out over progressively larger areas and peak flux decreases. For a conservative reflectivity of 0.9, peak flux is 4000 watts per square centimeter with zero error and about 3000 watts per square centimeter if the standard error is 2 milliradians (7 arc-min).

For various radial positions on the image plane, the values of flux ϕ in figure 2 were multiplied by $2\pi r$ and replotted in figure 3 in order to make graphic the selection of optimum aperture size. The significance of the ordinate in figure 3 stems from the following relation:

$$P = \int_0^R 2\pi r\phi \, dr$$

where

P thermal power entering aperture

R radius of aperture

r radius on image plane

Thus for any given aperture radius R, the area under any given solid line from 0 to R represents the solar power entering the aperture. In turn, the area under the same curve for all aperture radii greater than R represents the solar power striking the aperture plate and therefore lost by not entering the receiver cavity.

The heat radiated from the aperture itself is shown by the dashed lines in figure 3 for two values of cavity temperature. The value of 1200 K is characteristic of the maximum temperature of a number of power-conversion systems, and 1800 K is approximately the melting point of iron and thus is representative of high-temperature processing in space. The values of radiation from the aperture have been increased by 60 percent above the values for a blackbody in order to account for the thermal radiation from the aperture during the shadow portions as well as the sunlit portions of a low orbit about the Earth. Nominal values of 60 minutes of sunlight and 36 minutes of shadow were assumed. Thus, for both the solid and dashed lines the area under each line is proportional to the energies - for an entire orbit - that enter the aperture, that are reradiated through the aperture, or that are deflected by the aperture plate. The specific areas are identified in figure 4. For any given cavity temperature and given error in mirror surface, the image radius at which the dashed line crosses the solid line is the optimum aperture size. At radii smaller than the optimum, the solar flux exceeds the energy reradiated and at larger radii the reverse prevails.

For various given mirror errors, the net energy captured was integrated from zero to the optimum aperture radius. The resulting collection efficiencies are shown in figure 5. Collection efficiency is the ratio of the net energy captured to the solar energy incident upon the mirror; heat losses from the outer surface of the receiver were neglected. At low errors, efficiency asymptotically approaches the value of 0.9 assigned to mirror reflectivity. For a cavity temperature of 1200 K, collection efficiency is nearly constant for surface errors less than 1 milliradian (3 arc-min) and drops slowly to 0.8 for a standard error of 6 milliradians (21 arc-min). For a cavity temperature of 1800 K, collection efficiency is above 0.75 for mirror errors below 2 milliradians (7 arc-min). Thus, overall collection efficiencies over 0.80 are achievable, even at cavity temperatures as high as 1800 K, if only mirrors can be made with sufficient accuracy (1.5 mrad, or 5 arc-min).

Figure 6 shows a mirror 6 meters (20 ft) in diameter that was made by NASA Lewis of magnesium and in 12 sectors. Each sector was given its parabolic shape by creep-forming it over a heated, machined aluminum die. For each sector a plate of magnesium 2.5 centimeters thick was milled on the back in order to produce flanges along each edge and a roughly rectangular grid of ribs. The front surface and the ribs were all approximately 1.5 millimeters (60 mils) thick. After creep-forming, each sector was spray coated with epoxy. The surface tension of the epoxy produced a glossy surface onto which aluminum was deposited by vaporization in a vacuum. After the sectors were bolted together into a paraboloidal mirror, the mirror surface was inspected for accuracy by using the optical-inspection machine in figure 7. The standard deviation of the errors was about 1 milliradian (4 arc-min). The distribution of errors was also very close to a Gaussian curve, as had been assumed in analyzing the effects of mirror error on performance. This mirror weighed about 5 kilograms per square meter (1 lb/ft^2).

A mirror 1.8 meters (6 ft) in diameter was also made by NASA Lewis from 0.4-millimeter- (17-mil-) thick magnesium sheet, also by creep-forming the sectors over a heated aluminum form (fig. 8). The sectors were joined by slotted splines and epoxy (ref. 1). Total weight of the mirror was 1.6 kilograms per square meter (0.32 lb/ft^2), but its surface accuracy was not measured.

A comparable mirror was manufactured by TRW from aluminum sheet 0.4 millimeter (20 mils) thick by stretch-forming the sectors over a mandrel (ref. 2). Eight sectors and a rear supporting torus were bonded together into a paraboloid 1.5 meters (5 ft) in diameter. Just as for the Lewis mirrors, the front surface was coated with epoxy and aluminized. The standard deviation of errors in the mirror surface was 0.3 milliradian (1 arc-min). In full sunlight, such a mirror can supply over 900 watts per kilogram.

Thus, lightweight mirrors of sufficient accuracy for efficient collection of solar thermal energy (fig. 5), even if temperatures of about 1800 K are sought, have been built and tested on Earth. Thermal power outputs in excess of 1100 watts per square meter are achievable, a value 10 times the output presently attainable from solar arrays. Although these accurate mirrors have been assembled on Earth, large mirrors would require assembly or erection in space and this remains to be demonstrated. The heat from such mirrors can be used for power generation and/or space processing. For example, a paraboloid 100 meters in diameter appears potentially capable of supplying 5 megawatts of average thermal power in low Earth orbit. In most instances, the attainable temperature will be limited by the materials of the solar heat receiver rather than by the attainable accuracy of the concentrator.

NUCLEAR HEAT SOURCES

Either nuclear reactors or radioisotopes can also provide heat for direct use or for power generation. In terms of adaptation to the mission, nuclear energy sources are very different from solar sources. They are very compact and, since they operate with complete independence from the Sun, they permit operation in any Earth orbit without the constraint of orientation toward the Sun. In turn, operations in space can be significantly simplified because the field of view is not obscured, because rendezvous is simpler, and especially because the Earth (or any other celestial body) can be continuously observed without the constraint of also orienting an array of solar cells toward the Sun.

Life, cost, and nuclear-radiation shielding are all significant factors in nuclear heat sources. Plutonium-238 is the accepted radioisotope for spaceflights. Because its half-life is 87 years, the thermal output declines less than 8 percent in 10 years. Thus, the life of the radioisotope does not limit mission duration, in any practical sense. On the other hand, plutonium-238 costs about \$650 per watt of heat produced at the time the radioisotope capsules are manufactured. If radioisotopic decay is included, unit cost is roughly \$700 per thermal watt produced after 10 years. Obviously, this unit cost results in overall costs of \$700 000 per thermal kilowatt and \$700 million per thermal megawatt. Also, the total quantity of radioisotope that can be readily produced in a year's time is limited (ref. 3). These factors of unit cost and availability will make radioisotopic heat sources up to a few tens of kilowatts reasonable, but larger heat sources less reasonable. The technology for such radioisotopic heat sources is nearly all available, and a number of radioisotope power supplies have already been flown. The multihundred-watt capsules each produce 2400 watts of heat and operate at about 1100° C.

For nuclear reactors, reactor life is a design variable and very long lives (decades) are achievable. Basically, as heat is continually produced by the reactor, its fuel wears out. Two factors account for this wearing out:

(1) As uranium is progressively consumed, the reactor's ability to remain critical and to sustain a chain reaction declines.

(2) The fuel swells because of accumulating radiation damage to the fuel structure and because of accumulating fission products (2 product atoms for each uranium atom fissioned).

Within given limits on these two design variables, the reactor can be designed for almost any reasonable energy output, simply by incorporating enough fuel into the reactor and making the reactor large enough. Within the limit on energy output of a given reactor, power can be traded for life, and conversely.

As shown by reference 4, reactor weight increases fairly slowly if greater energy output (or longer life) is sought. Within a given family of reactors designed for the same operating temperature and a 7-year life, reactor weight is essentially constant for thermal powers from 16 to 200 kWt and increases only one-third as fast as thermal power from 200 to 1000 kWt. At the 1000-kWt level, reactor weight is estimated to be 360 kilograms. Similary, reactor cost will also change only slowly with power and life.

Reactor shield weight varies greatly with mission-related factors. For unmanned spacecraft, rather thin shields just between the reactor and payload (shadow shields) can be used and might weigh only a few hundred kilograms. On the other hand, even a shadow shield for manned flight might weigh 10 tons because of the low dose-rate limits specified for human beings. Such a shadow shield would prevent man's intrusion into the unshielded zone. Although uniform shielding all around the reactor (4π shielding) would give great operational freedom about the reactor, shield weight would then increase to perhaps 70 tons. Various shield weights between these limits can be achieved by compromising man's operational freedom about the reactor (chiefly with respect to solid angle) and by tailoring the reactor's shield design to fit these operational constraints. Thus, the reactor shield selected for manned flight will require detailed consideration of the relation between shield design and man's activities about the spacecraft. In contrast with shielding for reactors, radioisotopic heat sources using plutonium-238 require only minor shielding.

The radioisotope is most useful at low powers (below perhaps a few tens of kWt), and the reactor for high powers. The reasons for this stem from the facts (1) that the radioisotope, with its comparatively high unit cost, increases in cost in direct proportion to thermal power and (2) that the reactor and its shield increase only slowly in weight and cost as required thermal power increases. Thus, at high thermal powers, reactors would be the preferable nuclear heat source.

POWER-CONVERSION SYSTEMS

Some overall characteristics of thermal power-conversion systems are summarized in table I. Thermoelectric power systems have already flown as radioisotope thermoelectric generators (RTG's) on several long-lived spacecraft. To date, these power systems have produced powers up to 150 watts and had overall efficiencies of about 0.06. With modest advances in technology, efficiency up to perhaps 0.10 appears achievable. The RTG's are highly developed, rugged, inert in terms of interaction with a mission, and long lived. For long-duration missions that exploit the Space Transportation System, RTG's should be considered for both emergency power and free fliers.

Figure 9 shows, at the left, two multihundred-watt (MHW) RTG's mounted atop a Lincoln Experimental Satellite. The two RTG's produced 250 watts of electric power from two MHW heat sources. These same two MHW capsules are also intended for use in the mini-Brayton concept shown at the right. The Brayton conversion system, with its higher conversion efficiency, would produce

1300 watts from these same two highly developed heat sources. A 10-kilowatt version of such a Brayton power-conversion system has been under test for several years (fig. 10). The measured efficiency of this power-conversion system is 0.29 (fig. 11), but heat losses from a nuclear heat source might lower overall system efficiency to 0.27 or 0.28. Substituting already developed components would raise power-conversion-system efficiency to about 0.32 (ref. 4).

The main rotating component of this engine has a compressor, a turbine, and a generator on a single shaft supported by two gas-lubricated journal bearings and a double-acting thrust bearing (figs. 12 and 13). This rotating component has completed 36 000 hours of testing, and system performance has been stable over this period. Testing will continue toward a goal of 50 000 hours.

Organic Rankine systems for use in space have been investigated for about the past 15 years, and one power-conversion system operated for 8000 hours. Efficiency of 0.15 has been demonstrated, and 0.18 is projected for the future.

Thermionic converters have been investigated for generation of space power for about two decades. One converter operated stably for over 45 000 hours. Current concepts for thermionic powerplants (ref. 5) incorporate thermionic converters operating at 1650 K and having efficiencies of about 0.15. In reference 5, converter output at 0.15 efficiency is 500 kilowatts of unregulated power at 54 volts dc and about 9300 amperes. Power conditioning and regulation reduce this output to 343 kilowatts, for an overall efficiency of 0.10. Research on thermionic diodes (ref. 6) has shown that at higher temperatures (1800 to 2000 K) and high power densities (20 to 30 W/cm^2), converter efficiency can be raised to 0.3. In turn, overall system efficiency might then be about 0.2. As shown by reference 7, thermionic and reactor-Brayton powerplants for unmanned flight have about the same specific weight (g/W). However, exploitation of the higher power densities and higher operating temperatures of reference 6 will reduce thermionic weight significantly.

On the other hand, high operating temperatures cause more swelling of the reactor fuel or, alternatively, require reduced energy production from a given mass of fuel. In reference 7, UC-ZrC fuel is used for thermoelectric and Brayton power conversion within its swelling limits. Inasmuch as fuel swelling dominates reactor design in the megawatt range, the thermionic reactor could not use the UC-ZrC fuel within reasonable swelling limits. The solution was to switch to the more advanced Mo-UO_2 fuel and to increase reactor size. A larger reactor means increased shield weight, a crucial problem for manned flight inasmuch as shield weight dominates powerplant weight.

For a given reactor and shield operating at a given reactor temperature, a given amount of thermal energy can be produced over the mission. This energy might be used at a high rate but then only for a limited period. High power-conversion efficiency is an advantage because it would permit either the highest electrical power or the longest mission duration from a given reactor and shield. This is one of the outstanding characteristics of the Brayton power-conversion system. Among the power-conversion systems investigated for use in space, the Brayton system could also produce the largest electric power from any given heat source - be it solar mirror, radioisotope, or nuclear reactor.

MISSION INTEGRATION

Thermal power systems offer the opportunity for integrating the power system with the mission in ways very different from those offered by a photovoltaic power system. Factors of possible impact are as follows: (1) heat for processing in space, (2) heat for life support, (3) refrigeration of cryosensors, and (4) laser power. The Brayton power-conversion system is used as an example of what is achievable, chiefly because possible application of this system has received more attention.

Heat for Processing in Space

If heat for processing is needed at high temperature, some heat can be drawn directly from the power system's primary heat source. For nuclear heat sources, temperatures up to that of the heat source itself are usable. If still higher temperatures are needed, a solar mirror can readily achieve 2000 K with good efficiency (fig. 5). A solar mirror can provide about 1100 watts per square meter when in sunlight, about 10 times the value presently achievable with arrays of solar cells.

All the thermal power systems reject waste heat that might be useful to a mission. In a Brayton system optimized for high efficiency, this heat might be available in a fluid heated to 175° to 200° C (350° to 400° F). Higher temperatures are achievable with modest reductions in power-generation efficiency.

Heat for Life Support

Similarly, supplying heat for life support was investigated (ref. 8). In particular, adaptation of the Brayton cycle so as to provide heat at the required temperature was studied.

Consider now the problem of supplying equal amounts of energy in electrical and thermal forms, and compare area requirements for a photovoltaic and a solar-Brayton system in low Earth orbit (60 min of sunlight and 30 min of shade). Consider that the solar-Brayton system will continuously supply 10 kilowatts of electric power and 10 kilowatts of otherwise-rejected heat. For a collection efficiency of 0.8 and a conversion efficiency of 0.25, 54 square meters is required - corresponding to an average of 370 watts per square meter for a full sun-shade orbit. If for the photovoltaic system the combined efficiency of power processing and of battery charge and discharge is 0.7, if all the heat is produced during only the sunlit portion of the orbit, and if output of the photovoltaic array is taken as 140 watts per square meter (13 W/ft^2), 230 square meters is required - corresponding to an average output of 87 watts per square meter for a full sun-shade orbit. Thus, the photovoltaic array would require over four times the collector area of the solar-Brayton system. Requiring even modest amounts of heat at moderate temperature thus favors the thermal power systems.

Cryogenic Cooling

If a given spacecraft requires cryogenic cooling of, for example, infrared sensors for a long time, two reasonable choices are the Vuilleumier (VM) cooler and a Brayton system that is adapted for refrigeration as well as its usual function of producing power. In reference 9, the adapted Brayton system was analyzed and compared with the VM cooler.

The adapted Brayton cycle is shown in figure 14. The compressed gas is divided into two streams: one for power generation and the other for refrigeration. The refrigeration stream is then cooled in a radiator to the temperature at the compressor inlet. This stream is then further cooled in a recuperating heat exchanger. This compressed, cooled gas is then expanded in a turbine whose energy extraction further cools the gas and whose power output augments that of the Brayton power system. The cold gas at the turbine discharge provides the cryogenic cooling and is then reheated in the recuperating heat exchanger almost back to the compressor inlet temperature.

For this cooling application, the Brayton system is operated on neon gas, which liquefies at approximately 27 K. For the analysis in reference 9, cooling by the Brayton system was limited to 50 K in order to avoid any liquefaction and thereby to simplify the performance calculations.

A cooling load of 40 thermal watts at 50 K was selected. For this cooling load, a VM cooler continuously requires about 120 watts of electric power and 3200 watts of heat. A photovoltaic array is assumed to produce electric power for both the power and heat demands. The orbital period is taken as 90 minutes and the sunlit portion as 60 minutes. Charge-discharge efficiency of the batteries is taken as 0.7 and the array output as 140 watts per square meter. For these conditions, the required array area is 41 square meters. Reference 9 shows that the refrigeration load reduces the electric power output of the Brayton system by 700 watts. If the efficiency of the Brayton power-conversion system is taken as 0.3, the collection efficiency as 0.8, and the sunlit period as 60 minutes in a 90-minute orbit, the added collector area required in order to regain the 700 watts of electric power is 3.1 square meters. Thus, the photovoltaic system requires 13 times as much collection area as the Brayton system.

Laser Power Transmission

A concept for generation of a gas-laser beam by adaptation of a Brayton power system was analyzed in reference 10. The concept is shown schematically in figure 15. Gas for operation of the laser is first compressed and then heated in the recuperator and in the nuclear reactor. After rapid expansion in a supersonic nozzle, the gas in temporary disequilibrium emits its beam of laser power. The resulting high-velocity stream is diffused in order that much of its kinetic energy might be recovered. The resulting stream of still-hot gas then passes through a turbine that drives the compressor and an alternator.

SUMMARY OF RESULTS

In considering space power generation systems as alternatives to photovoltaic systems, the following conclusions have been reached:

1. Radioisotope thermoelectric generators (RTG's) are highly developed and available for orbital application. Their inherently low powers make them most appropriate for special applications such as emergency power or free fliers.

2. Solar paraboloidal mirrors are suitable for supplying heat for either power generation or space processing. Mirror accuracy already demonstrated is sufficient for temperatures required to melt iron (1800 K). However, the means for assembling or erecting these mirrors in space are not yet developed. Collector areas required to supply process heat are only 1/10 those required by arrays of solar cells.

3. Dynamic power systems can use heat from either solar or nuclear sources. The highest efficiency and longest life have been demonstrated by the Brayton system, which has so far attained efficiency of over 25 percent and life in excess of 4 years.

4. The thermal power systems provide unusual opportunities in mission integration. A given solar mirror might drive a thermal power system as well as provide high-temperature heat for space processing. Otherwise-wasted heat from power generation can be used for life support, and the Brayton system can also be adapted for cryo-cooling of infrared sensors. If intense beams are needed from gas-dynamic lasers, the Brayton system might not only provide the hot, pressurized gas for the laser but also produce electric power from the hot gas stream exhausted by the laser.

REFERENCES

1. Richter, Carl W.; Birchenough, Arthur G.; et al.: Design and Fabrication of a Low-Specific-Weight Parabolic Dish Solar Concentrator. NASA TP-1152, 1978.

2. Heath, Atwood R.; and Hoffman, Edward L.: Recent Gains in Solar Concentrator Technology. J. Spacecr. Rockets, vol. 4, no. 5, May 1967, pp. 621-624.

3. Ad Hoc Study Group: Selection of Radioisotopes for Space Power Systems. NASA TM X-1212, 1966.

4. Klann, John L.; and Wintucky, William T.: Status of the 2 to 15 kWe Brayton Power System and Potential Gains from Component Improvements. NASA TM X-67835, 1971. (Also IECEC paper.)

5. Pawlik, Eugene V.; and Phillips, Wayne M.: A Nuclear Electric Propulsion Vehicle for Planetary Exploration. J. Spacecraft, vol. 14, 1977, pp. 518-525.

6. Morris, James F.. High-Temperature, High-Power-Density Thermionic Energy Conversion for Space. NASA TM X-73844, 1977.

7. Koenig, Daniel R.; Ranken, William A.; and Salmi, Ernest W.: Heat-Pipe Reactors for Space Power Applications. J. Energy, vol. 1, 1977, pp. 237-243.

8. Klann, John L.. Thermal Integration of a Brayton Power System with a Life-Support System. NASA TM X-1965, 1970.

9. Klann, John L.. Analysis of a Combined Refrigerator-Generator Space Power System. NASA TM X-71433, 1973.

10. Burns, Raymond K.. Parametric Thermodynamic Analysis of a Closed-Cycle Gas-Laser Operation in Space. NASA TN D-7658, 1974.

TABLE I. - ENERGY CONVERSION CHARACTERISTICS

Concept	Present efficiency	Future efficiency	Demonstrated endurance, hr
Thermoelectric	0.06	0.10	(a)
Organic Rankine	.15	.18	8 000
Brayton	.25	.32	36 000
Thermionic	.10	.20	45 000

[a]Spaceflight.

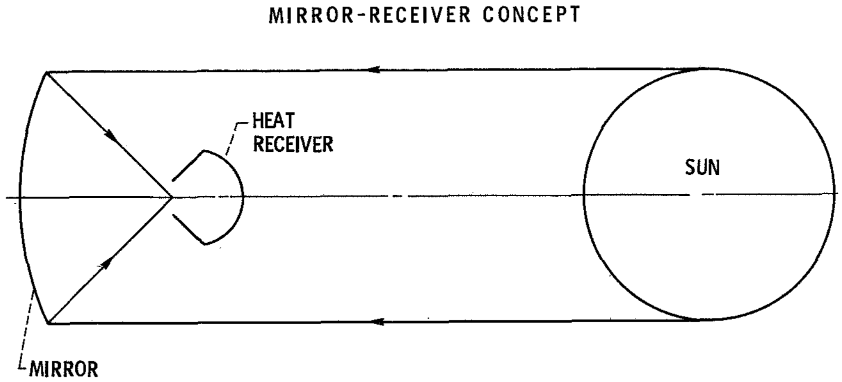


Figure 1.

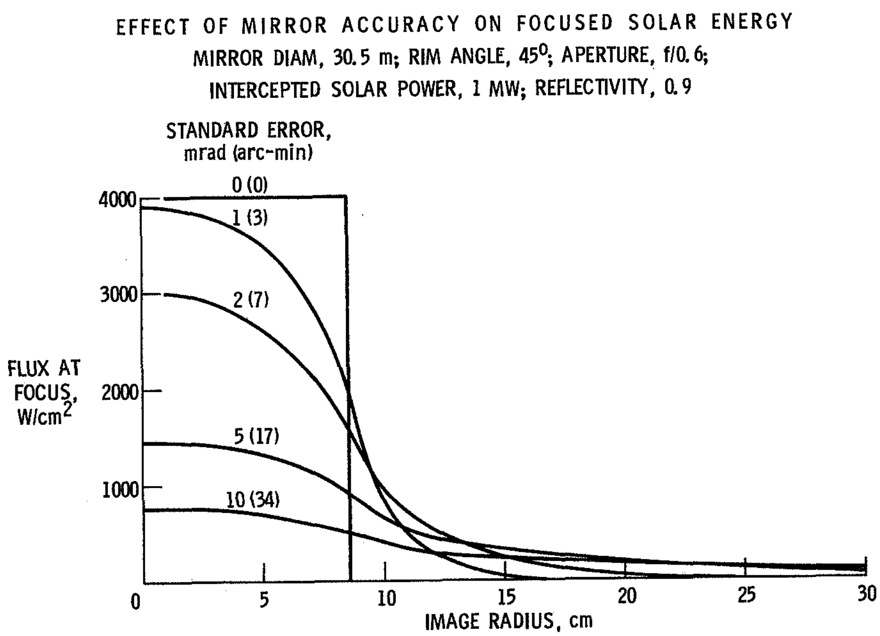


Figure 2.

MIRROR ACCURACY AND ENERGY COLLECTION:
INCIDENT AND RERADIATED ENERGIES
60 min OF SUNLIGHT IN 96-min ORBIT; MIRROR DIAM, 30.5 m

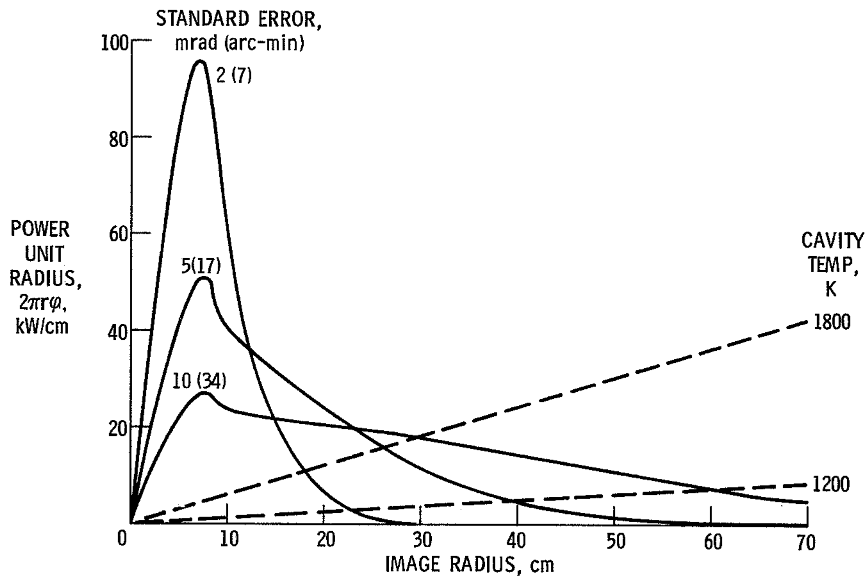


Figure 3.

ENERGIES CAPTURED, RERADIATED AND LOST
STANDARD ERROR, 10 mrad

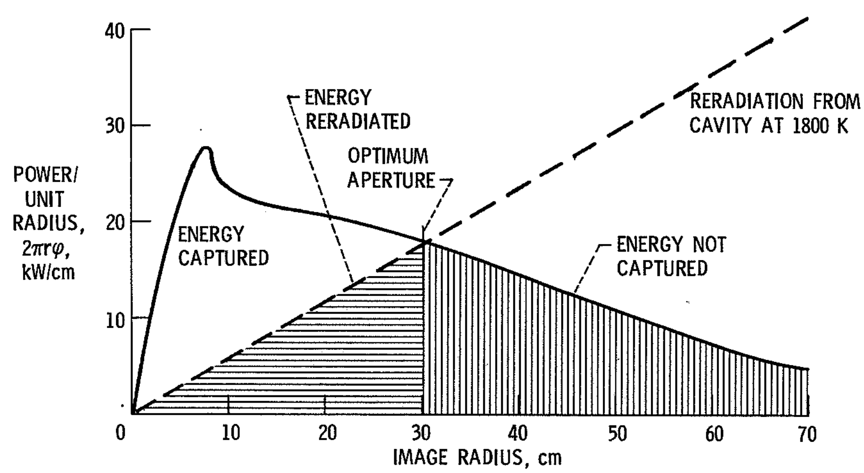


Figure 4.

EFFICIENCY OF COLLECTING SOLAR ENERGY

ORBIT, 96 min; SUNLIT PORTION, 60 min; REFLECTIVITY, 0.9

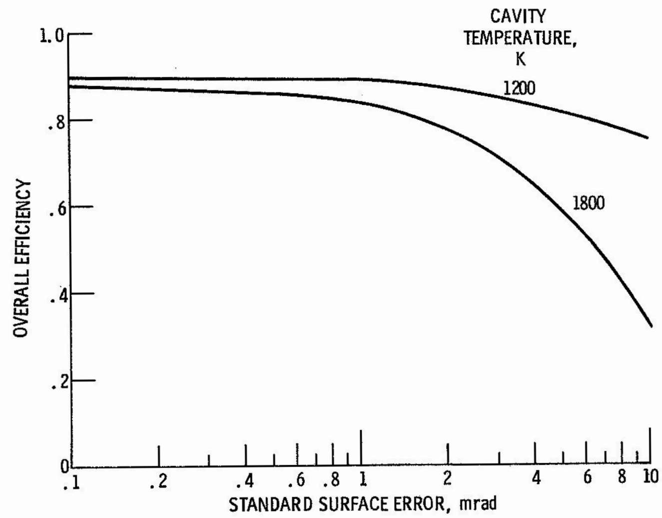


Figure 5.

SOLAR MIRROR - 20 ft IN DIAM

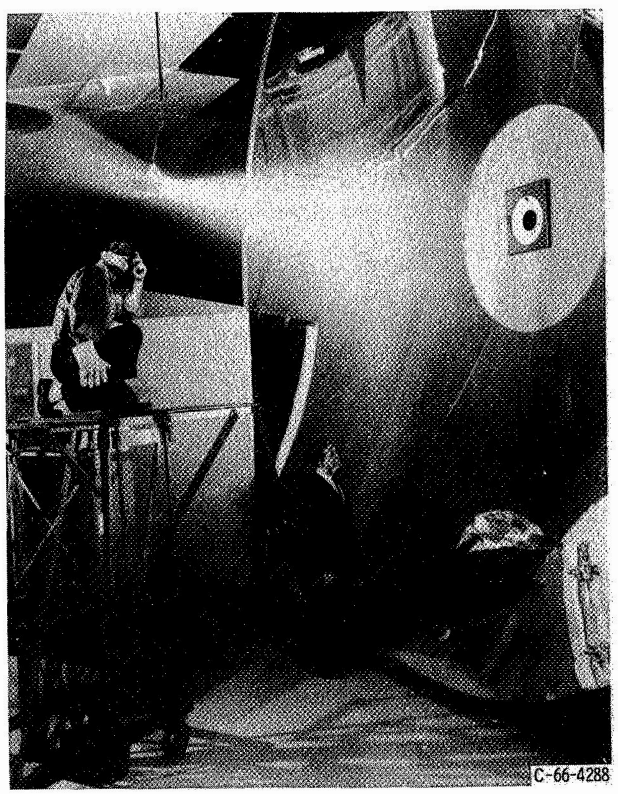


Figure 6.

MIRROR INSPECTION MACHINE

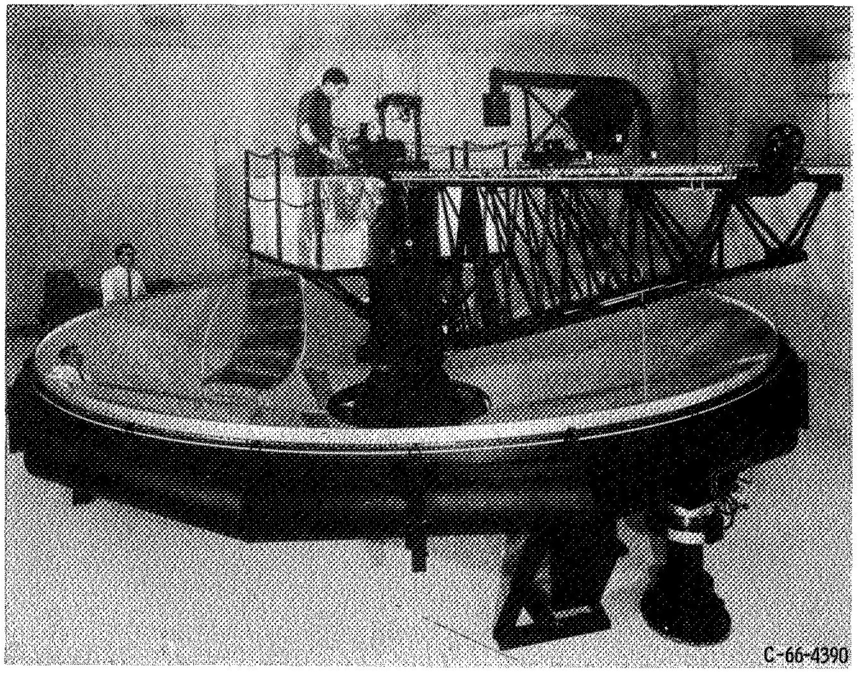


Figure 7.

SOLAR MIRROR - 6 ft IN DIAM

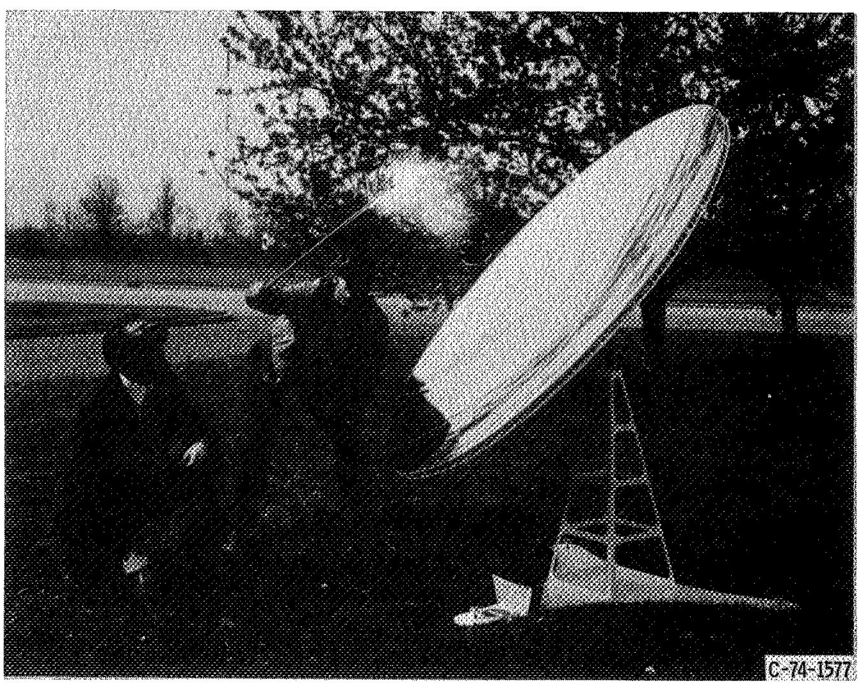


Figure 8.

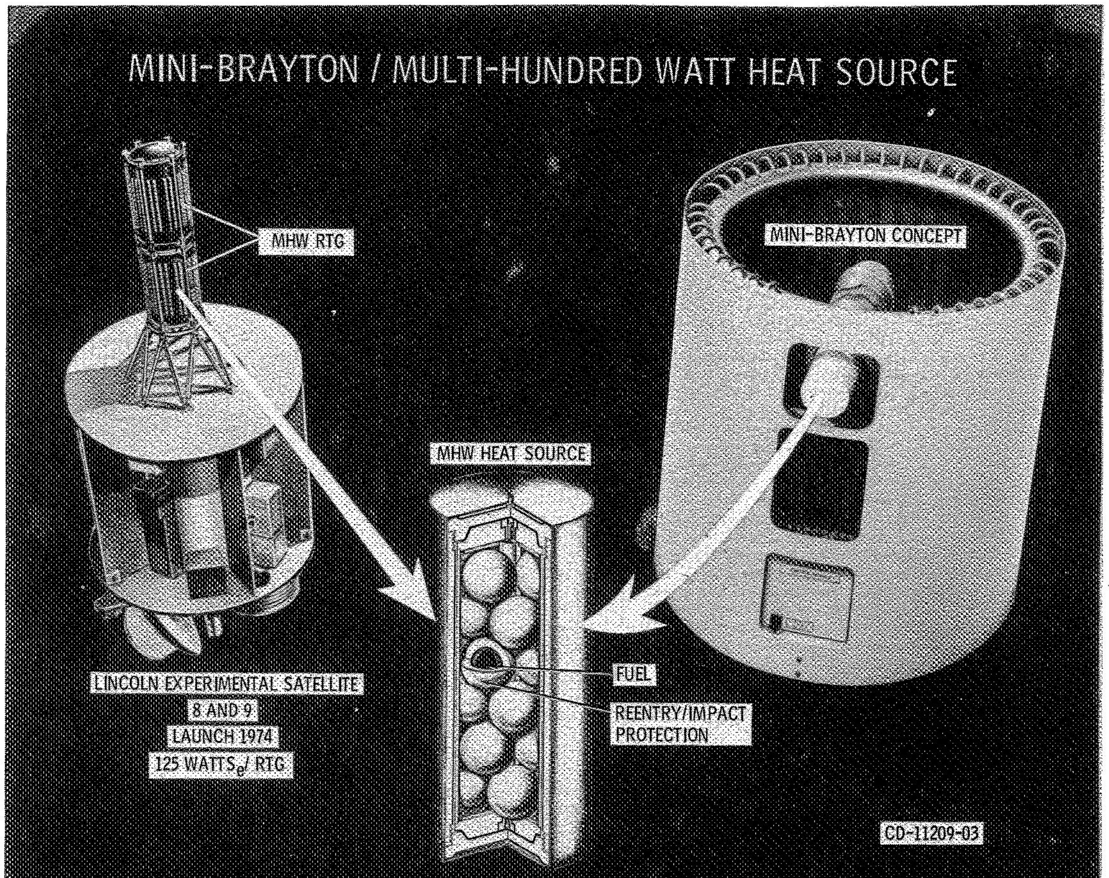


Figure 9.

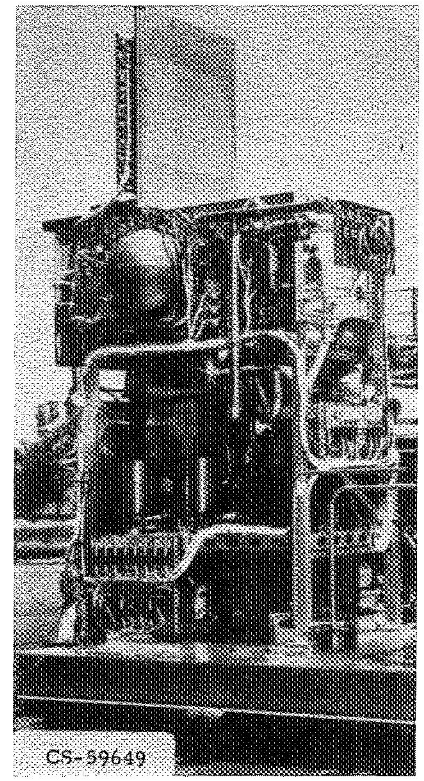


Figure 10.

BRAYTON ENGINE PERFORMANCE

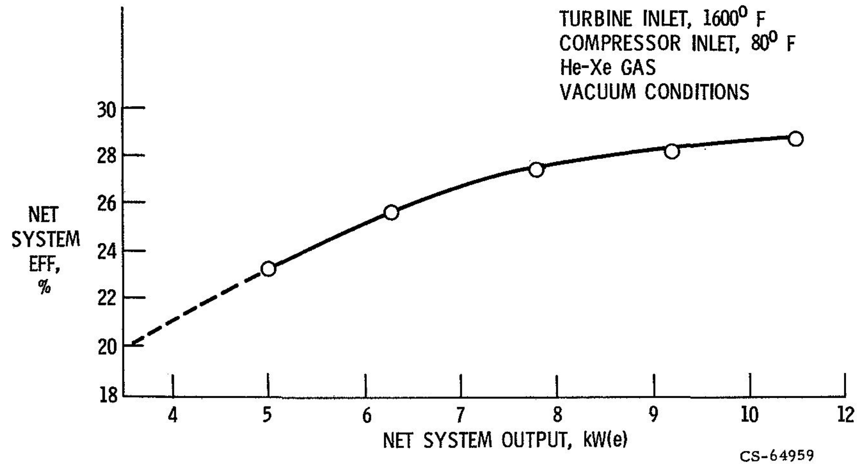


Figure 11.

BRAYTON ROTATING UNIT

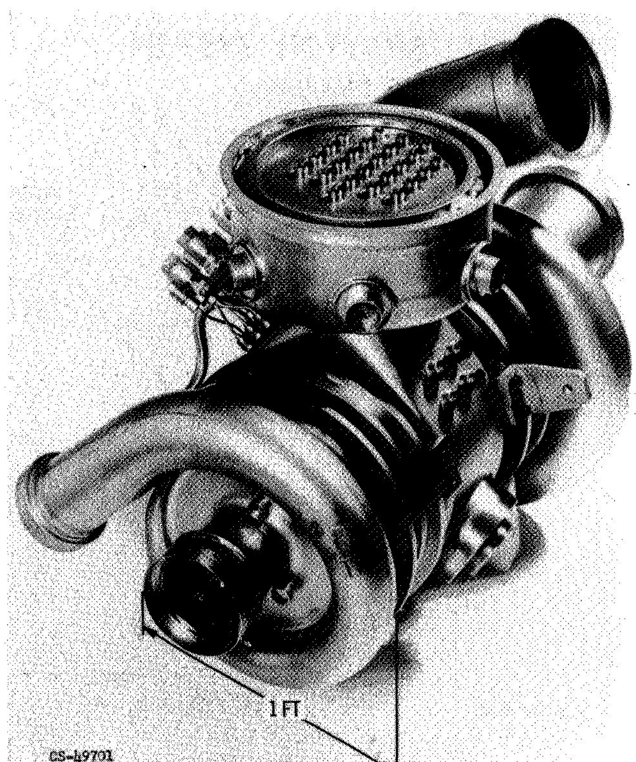
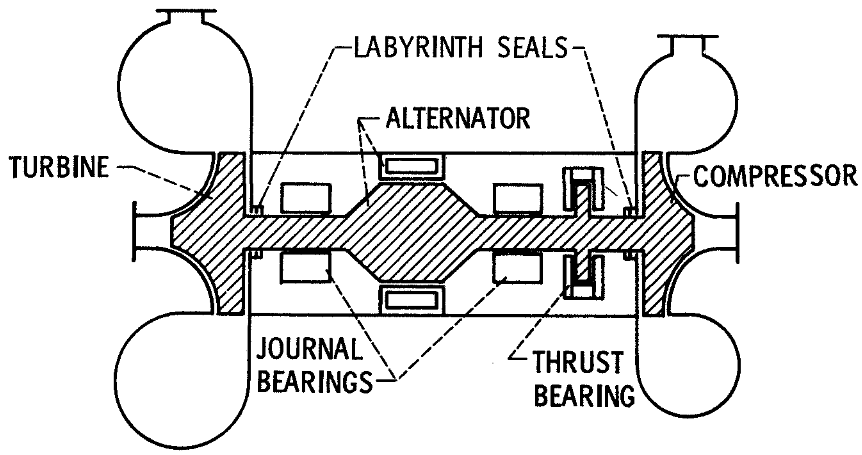


Figure 12.

BRU SCHEMATIC

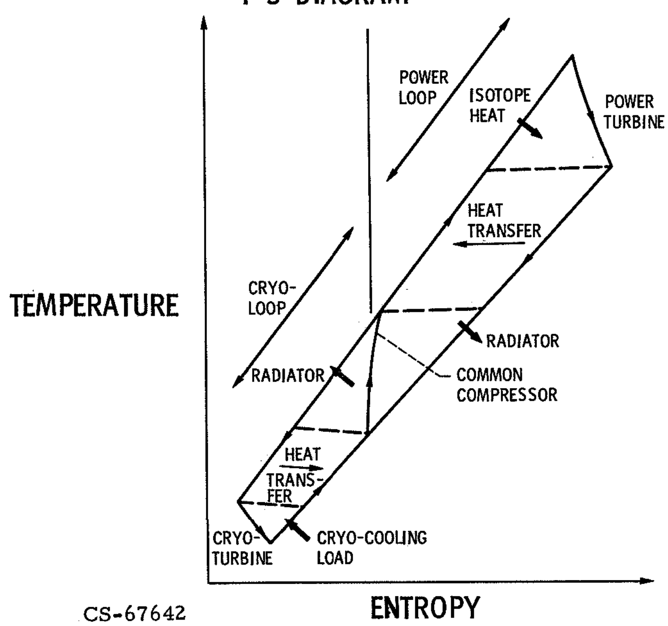


CS-55418

Figure 13.

DUAL BRAYTON SYSTEM

T-S DIAGRAM



CS-67642

Figure 14.

CONCEPT FOR A LASER–BRAYTON POWER SYSTEM

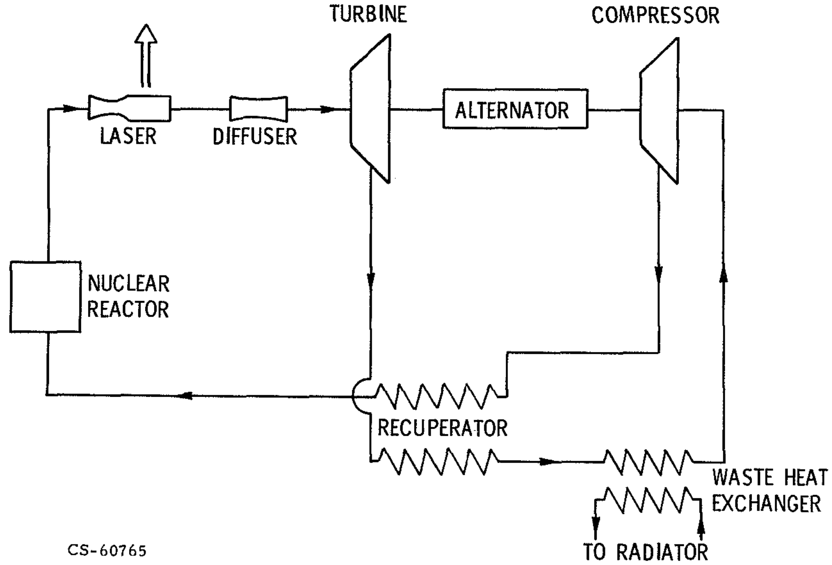


Figure 15.